N63-15055

15p

NASA TN D-1747

NASA TN D-1747

Code 1

TECHNICAL NOTE

D-1747

VEHICLE TECHNOLOGY CONSIDERATIONS FOR A SOLAR PROBE

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

WASHINGTON April 1963

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SUMMARY

This report demonstrates that solar probes are feasible from the viewpoint of presently planned vehicle technology. For the early attempts the Saturn C-1 is considered to be the most likely vehicle to place the final stage or stages into a circular orbit. If two slightly modified but identical Centaur Jr. vehicles are used for the upper stages, then this four stage configuration will be capable of placing from 400 to 2500 pounds at perihelion distances of 0.120 to 0.245 A. U. A more reliable three stage configuration, using one Centaur Jr. for the upper stage but requiring greater modification in the original design of the vehicle, would place payloads of 400 to 2500 pounds at perihelion distances of 0.185 to 0.290 A. U.

Analysis of the upper stages was based on the payload value obtained for the Saturn vehicle by using the "Generalized Powered Flight Trajectory Program". At burnout there would be a payload of 24,612 pounds in a 100 nautical mile parking orbit.

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INTRODUCTION

This report continues the investigation and analysis of a possible solar probe attempt in this decade by extending the studies of References 1 and 2. Reference 1 includes temperature and solar radiation pressure considerations, and analytical and numerical trajectory studies. Reference 2 presents velocity requirements for achieving desired perihelion distances for both one- and two-impulse orbits, and concludes from time considerations that the one-impulse method is the more advantageous. Probe weight for various perihelion distances is given for two configurations, a Saturn C-1 plus H_2-F_2 stages and an Atlas-Centaur plus a solid stage.

Other vehicle configurations, using O_2-H_2 stages presently in existence or under consideration, which can place usable payloads of 400 to 2500 pounds at perihelion distances of 0.1 to 0.3 A.U. are investigated in the current study. After considering the probe's trajectory and velocity requirements, we will use them to determine the number of stages necessary. This report also examines future vehicle payload capabilities and specific impulse demands for the upper stages. For the first attempt, the lightest structure considered – the Saturn C-1 – appears to be the most likely vehicle to place the required payload in a circular orbit. The trajectory of the C-1 rocket was obtained from the IBM 7090 computer version of the "Generalized Powered Flight Trajectory Program" developed at the Jet Propulsion Laboratory and is included here.

For all upper stage designs, specific impulses of 420 seconds and thrusts of 15,000 pounds have been specified. The upper stage designs presented here emphasize: (1) payload optimization, achieved when each stage (two upper stages) acquires equal velocity increments; (2) design practicability, by having one design for each of two upper stages; (3) maximum reliability, by using, for example, a three stage rocket for less ambitious missions. Because of structural requirements payload optimization cannot be achieved, but the latter two aims can be, the more appropriate depending on a mission's goal and the payload specifications. The results of this report demonstrate that a solar probe is possible from the viewpoint of presently planned vehicle technology and its purpose is to generate interest in the solar probe effort.

Experimental areas for solar probes include, as stated by Krafft A. Ehricke (Reference 3, pages 107 and 108),

> the nature of the solar corona (ionization and excitation of coronal atoms); extension of coronal material into space; support and heating of coronal material; electron density in inner and outer corona; electric and magnetic fields in near-solar space; evaporation of protons and electrons from the corona and solar corpuscular radiation in general; solar cosmic radiation and extension of the solar atmosphere into interplanetary space.

Ehricke also mentions that "Another intriguing mission of the solar probe is the exploration of the zodiacal light. . . ." He points out that very little is known about the composition, mass, and motion of the huge disc-shaped cloud that causes this light. The entire scientific significance of solar probes cannot be appropriately assessed.

VEHICLE VELOCITY REQUIREMENT VS. PERIHELION DISTANCE*

The velocity v needed with respect to the earth, as a function of the desired perihelion distance r_π, is derived in this section. Let the orbital plane of the vehicle be in the plane of the ecliptic (Figure 1), i.e.,

$$\frac{\mathbf{v}_\infty \cdot \mathbf{U}}{|\mathbf{v}_\infty \cdot \mathbf{U}|} = +1 ,$$

where $\mathbf{U}$ is the earth's orbital velocity, $\mathbf{v}_\infty = \mathbf{U} - \mathbf{V}_1$, and $\mathbf{V}_1$ is the heliocentric departure velocity, i.e., the aphelion velocity for the ellipse. If the departure is at the aphelion of the transfer ellipse, then

$$V_1 = \sqrt{\frac{2K_\odot r_\pi}{r_A (r_\pi + r_A)}} , \qquad (1)$$

where $K_\odot$ is the gravitational constant for the sun, $r_\pi = m r_A$ is the perihelion distance, $0 < m < 1$, and r_A, the aphelion distance, is 1 A.U. The hyperbolic departure velocity with respect to the earth is

$$v = \sqrt{\frac{2K}{r} + v_\infty^2} , \qquad (2)$$

where K is the gravitational constant for the earth, r is the distance of the vehicle from the earth at the time of injection, and $2K/r$ is the escape velocity from the earth. By substituting $v_\infty = U - V_1$, $K_\odot/r_A = U^2$, $r_A = 1$ A.U., and Equation 1 into Equation 2, we have

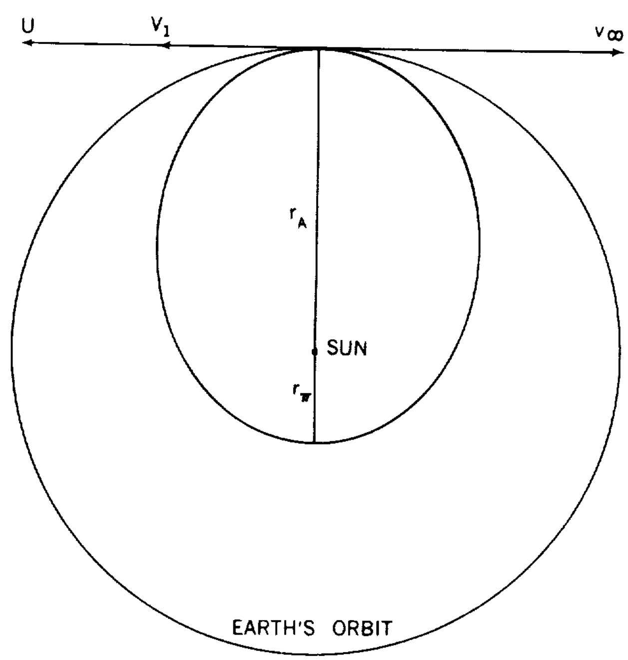

Figure 1—Solar probe trajectory.

*The equations used in this section have been developed from Reference 4.

$$v = \sqrt{\frac{2K}{r} + U^2 \left(1 - \sqrt{\frac{2 r_\pi}{1 + r_\pi}}\right)^2} \, , \tag{3}$$

where U is the earth's circular velocity about the sun and in astronomical units $0 \leq r_\pi \leq 1$.

TRANSIT TIME

If P_E is the period for the earth's motion around the sun and P_v is the period for the vehicle's motion around the sun, then

$$\frac{P_v}{P_E} = \frac{a_v^{3/2}}{a_E^{3/2}} = \frac{\left(\frac{r_A + r_\pi}{2}\right)^{3/2}}{r_A^{3/2}} \, ,$$

where a_v is the semimajor axis of the transfer ellipse and a_E is the semimajor axis of the earth's orbit. By substituting r_A = 1 A.U. and P_E = 365.25 days we find

$$T = \frac{P_v}{2} = 64.5677 \, (1 + r_\pi)^{3/2} \text{ days,} \tag{4}$$

the transit time of the vehicle from aphelion to perihelion. The expressions $v = f(r_\pi)$ for Equation 3 and $T = g(r_\pi)$ for Equation 4 are graphically presented in Figures 2 and 3, respectively, where $0 \geq r_\pi \geq 0.3$. In Figure 12 in Reference 1 both velocity and transit time vs. perihelion distance for the vehicle are shown with the earth at perihelion and at aphelion. Figures 2 and 3 in this report represent the same functions for the earth at 1 A.U., and they agree with the average of the two curves in Reference 1 to within 0.1 km/sec.

CHEMICAL ROCKET ANALYSIS

The well-known rocket equation is studied in considering such factors as:

1. The minimum number of stages necessary to attain the required velocity;
2. The velocity cut-off value which would optimize the final payload;
3. Possible vehicles for placing the upper stages into a parking orbit, from the viewpoints of present and future vehicle technology.

The rocket equation is:

$$\Delta v \quad g_0 \, I_s \ln \left(\frac{M_0}{M_f}\right) , \tag{5}$$

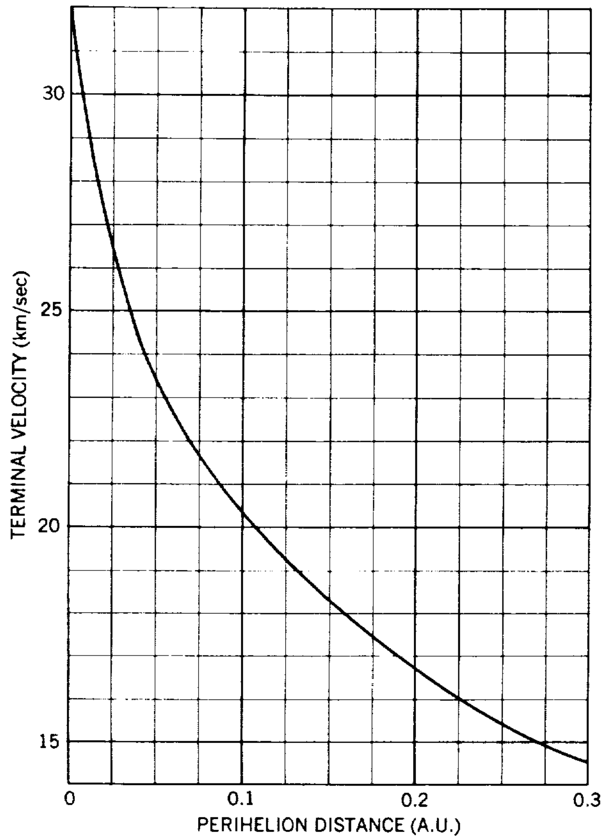

Figure 2—Terminal velocity vs. perihelion distance.

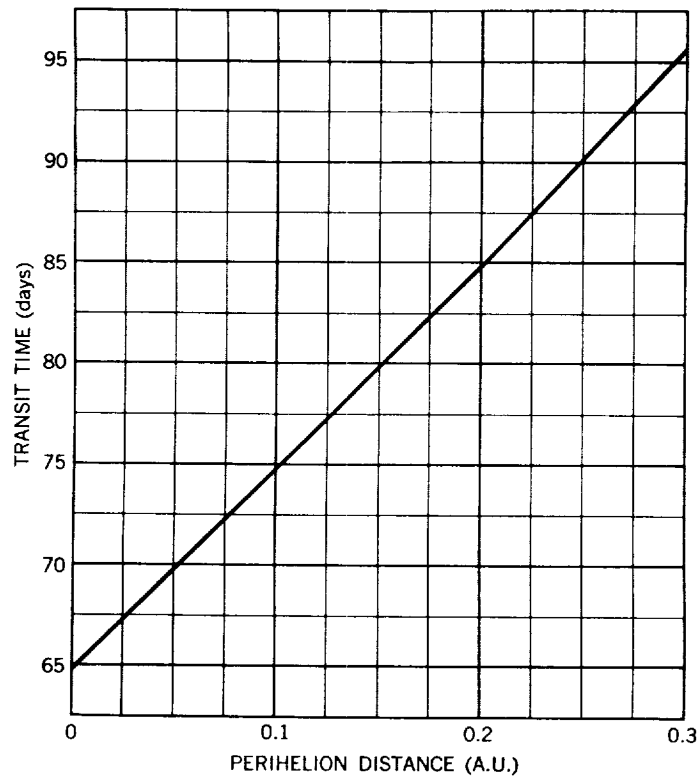

Figure 3—Vehicle's transit time from earth to perihelion vs. perihelion distance.

where Δv is the increment of velocity from epoch to burnout, g_o is the gravitational constant, I_s is the specific impulse, M_0 is the initial weight of the vehicle, and M_f is the weight at burnout. Let M_p be the propellant weight, M_s be the structural weight ($0.1\ M_p$), M_l be the payload weight, and

$$x = \frac{\Delta v}{g_o\ I_s} \quad ;$$

then the ideal equation for chemical rockets becomes

$$e^x = \frac{M_p + M_s + M_l}{M_s + M_l} \quad . \tag{6}$$

Substituting $M_s = 0.1\ M_p$ and $M_0 = M_p + M_s + M_l$ into Equation 6, we find

$$\frac{M_0}{M_l} = \frac{10\ e^x}{11 - e^x} \quad , \tag{7}$$

and we see that $M_l > 0$ for $e^x < 11$ or $x < 2.4$. If the specific impulse is 422 seconds and $r_\pi = 0.1$ A.U., then $v = 20.33$ km, from Equation 3. We are here concerned with the velocity increment after circular orbit velocity is achieved; the total velocity increment for the upper stages is $\Delta v = v - v_a$, where v_a

is the circular velocity, 7.8 km/sec. Substitution of the above values gives Δv = (20.33 - 7.8) km/sec = 12.53 km/sec, and from the definition of x,

$$x = \frac{12.53}{4.1356} = 3.03 > 2.4 .$$

Thus, we must have at least two stages after the vehicle is in a parking orbit in order that $\Delta v = 12.53$ km/sec. Conversely, the minimum perihelion we can obtain with a single stage vehicle is 0.167 A.U. for $M_l = 0$. The next problem is to determine the velocity cut-off of the first stage of the two stage rocket (after achieving the parking orbit) which would maximize the final payload. Let M_c be the weight in the parking orbit, M_i the weight of the last stage plus the payload, k the fractional part of Δv for the first stage, and c the fractional part of Δv for the second stage $(k + c = 1)$. From Equation 7,

$$M_i = M_l \frac{10\, e^{cx}}{11 - e^{cx}} , \tag{8}$$

$$M_c = M_i \frac{10\, e^{kx}}{11 - e^{kx}} . \tag{9}$$

Substituting Equation 8 into Equation 9, we have

$$M_c = M_l \left(\frac{10\, e^{cx}}{11 - e^{cx}}\right)\left(\frac{10\, e^{kx}}{11 - e^{kx}}\right) . \tag{10}$$

Replacing c by $1 - k$ in Equation 10 and simplifying yields

$$\frac{M_c}{M_l} = \frac{100\, e^{x}\, e^{kx}}{121\, e^{kx} - 11\, e^{x} - 11\, e^{2kx} + e^{x}\, e^{kx}} \tag{11}$$

We wish to differentiate Equation 11 with respect to k in order to find what value of k will give M_c/M_l its minimum value. Performing the differentiation and equating the result to zero,

$$\frac{d}{dk}\left(\frac{M_c}{M_l}\right) = 0 ,$$

shows that M_c/M_l is a minimum when $k = 1/2$. In this case $k = 1/2 = c$ and Equation 10 becomes

$$\frac{M_c}{M_l} = \left(\frac{10\, e^{x/2}}{11 - e^{x/2}}\right)^2 . \tag{12}$$

With

$$x = \frac{\Delta v}{g_o\, I_s} = \frac{12.53}{g_o\, I_s} ,$$

and M_l = 500 pounds, a reasonable payload, the specified impulse was varied and the graph for $M_c = h(I_s)$ was derived as shown in Figure 4. The last problem mentioned at the beginning of this section, selecting a vehicle which will place the upper stages into a circular orbit, is resolved by examining Figure 4 which shows that at least 25,000 pounds is required in a parking orbit for the maximum specific impulse being considered. Since the orbiting of payloads heavier than 25,000 pounds at present seems more distant than designing engines with specific impulses of 420 seconds, the C-1 vehicle has been chosen, its circular orbit payload capability being the required 25,000 pounds.

PAYLOAD IN A CIRCULAR ORBIT

The Saturn C-1 vehicle was programmed for a 100 nautical mile circular orbit with the "Generalized Powered Flight Trajectory Program" developed by the Jet Propulsion Laboratory. For 15 seconds the vehicle followed a vertical path. It was then tilted at a rate of 0.7877 deg/sec for 15 seconds. The remainder of the first stage was programmed with zero lift. A pitch up control of 13 degrees above the local horizon was necessary during the second stage in order for the vehicle to enter into a circular orbit with zero flight path angle. The results for the trajectory are shown in the Figures 5-9. Payload weight in the circular orbit is 24,612 pounds.

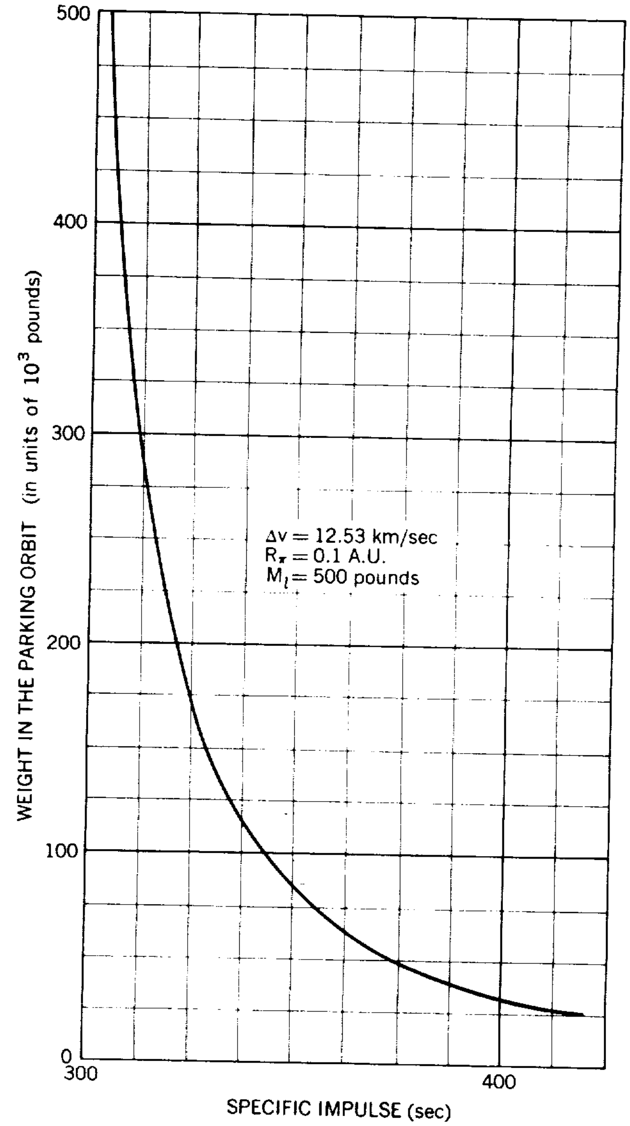

Figure 4—Weight in a parking orbit vs. the specific impulse.

UPPER STAGE DESIGNS

Three upper stage designs will be presented:

1. The design in which each stage acquires equal velocity increments. This design optimizes the payload, as discussed earlier.
2. One design for both upper stages, facilitating the engineering and lowering the cost of the project.
3. The design of a three stage rocket for less ambitious, more reliable missions.

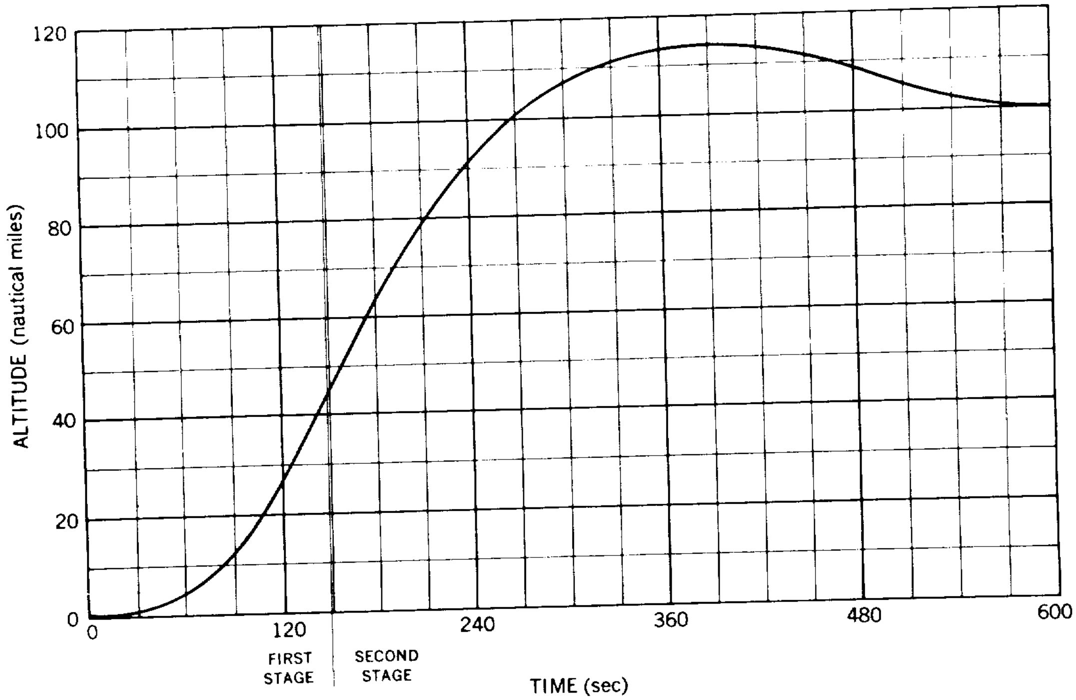

Figure 5—Altitude vs. time for the Saturn C-1 trajectory for a 100 nautical mile parking orbit.

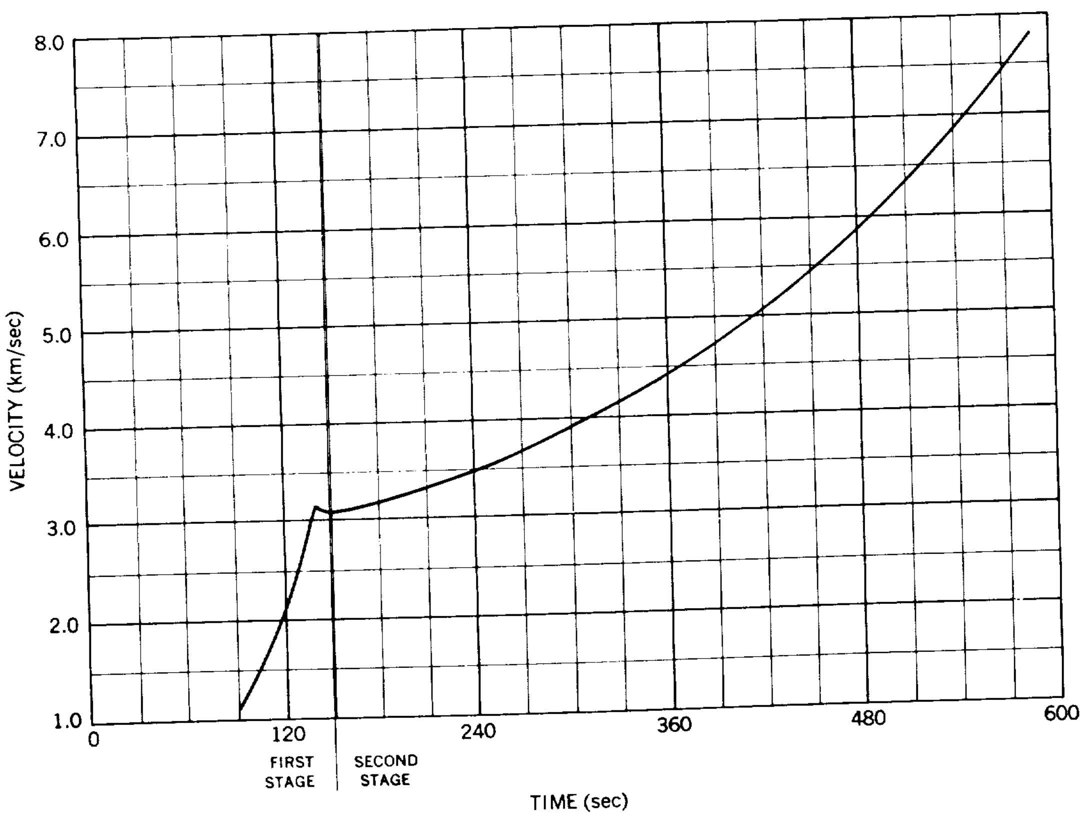

Figure 6—Velocity vs. time for the Saturn C-1 trajectory for a 100 nautical mile parking orbit.

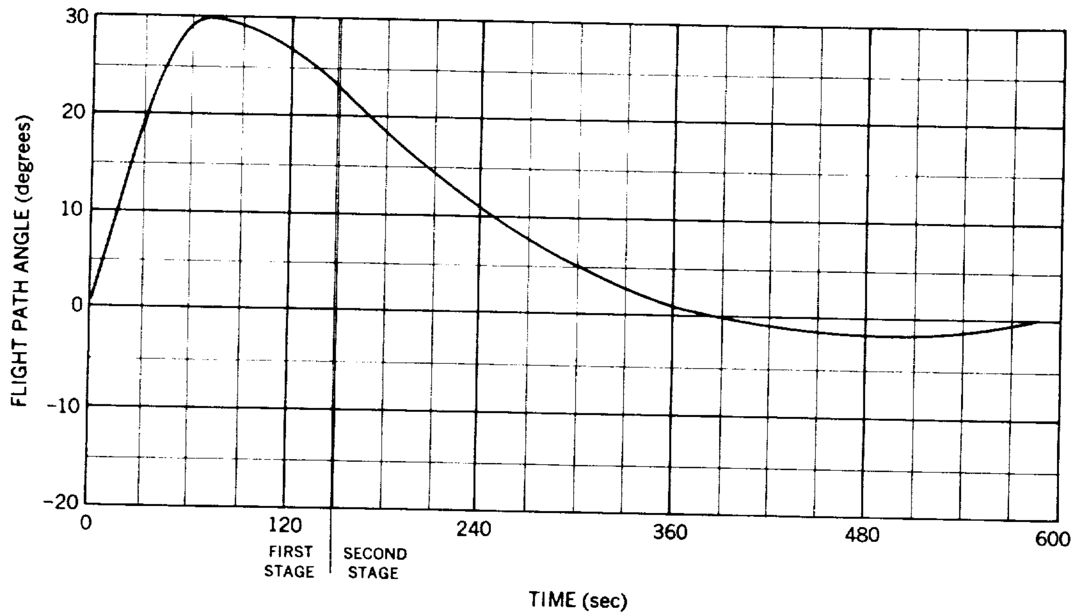

Figure 7—Flight path angle vs. time for the Saturn C-1 trajectory for a 100 nautical mile parking orbit.

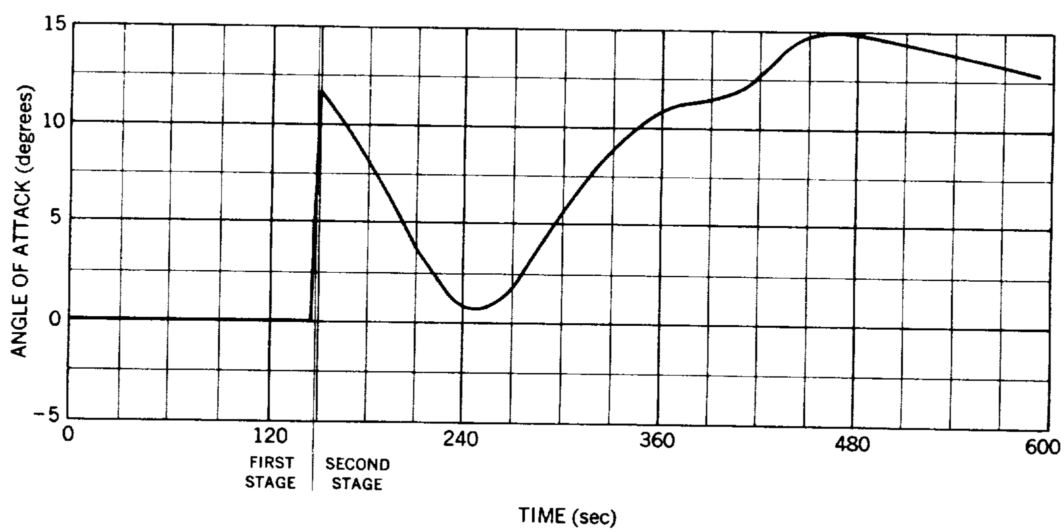

Figure 8—Angle of attack vs. time for the Saturn C-1 trajectory for a 100 nautical mile parking orbit.

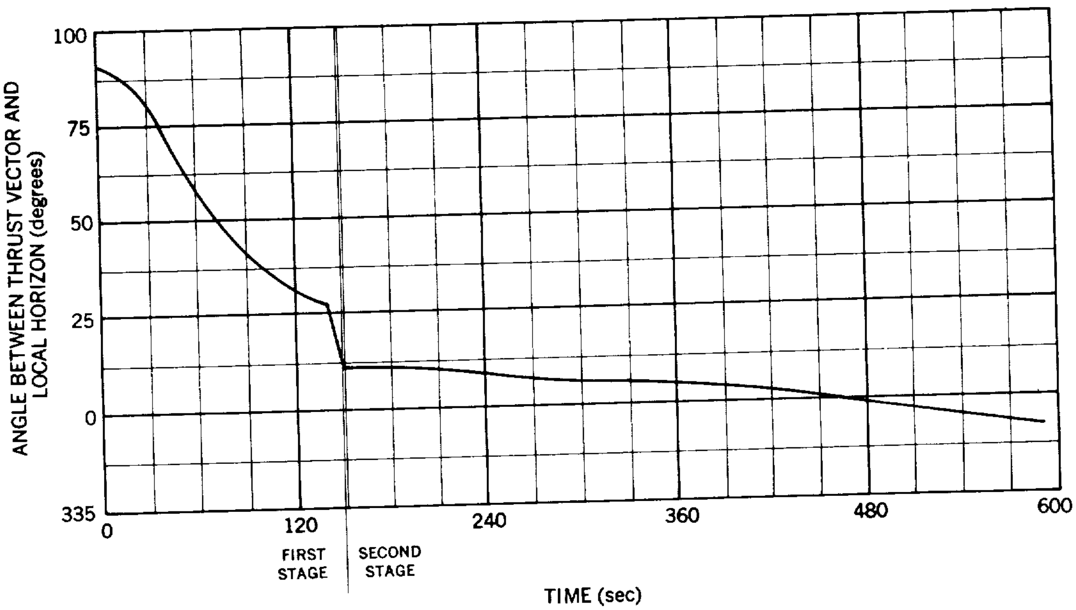

Figure 9—Angle between thrust vector and local horizon vs. time for the Saturn C-1 trajectory for a 100 nautical mile parking orbit.

Optimum Final Payload

For equal velocity increments in each stage let $k = 1/2 = c$. Since the total velocity is $\Delta v = \Delta v_1 + \Delta v_2 = 12.53$ km/sec, then $\Delta v_1 = 6.265$ km/sec $= \Delta v_2$. From Equation 9, for $M_c = 24{,}612$ pounds, $k = 1/2$, and

$$x = \frac{\Delta v_1}{g_o I_s} ,$$

where $I_s = 422$ sec, we find that $M_i = 3445$ pounds. In Equation 8 substitute $M_i = 3445$, $c = 1/2$, and

$$x = \frac{\Delta v_2}{g_o I_s} ,$$

then $M_l = 482$ pounds. Now $M_i = M_{p4} + M_{s4} + M_l$, where M_i is the total weight of the fourth stage, $M_{p4} = 10 M_{s4}$, and the number "4" indicates that the value applies to the fourth stage. Therefore $M_i = 11 M_{s4} + M_l$, and the structural weight for the fourth stage is

$$M_{s4} = \frac{M_i - M_l}{11} = 270 \text{ pounds} .$$

The required structural weight of 270 pounds is not reasonable for the engine we are considering, a more reasonable estimate, taken from vehicle specifications, being 1100 pounds. At 0.2 A.U., payload optimization is satisfied when the structural weight for the fourth stage is less than 445 pounds. Therefore it is necessary to deviate from the attempt of attaining equal velocities for each stage.

One Design for the Two Upper Stages

We will now investigate the possibility of using one design for two upper stages, each of which has a structural weight of 1096 pounds, a reasonable figure for an engine such as the Centaur Jr. Let M_c be 24,612 pounds, M_s be 1096 pounds, I_s be 420 seconds, M_{p3}, the propellant weight of the third stage = $10M_s$ = 10,960 pounds, and M_{p4}, the propellant weight of the fourth stage = 11,460 - M_l. Substituting these values into the rocket equation yields

$$\exp \frac{\Delta v_1}{4.116} = \frac{M_c}{M_s + M_{p4} + M_l} = \frac{24,612}{13,652} , \tag{13}$$

or

$$\Delta v_1 = 2.6943 \text{ km/sec}.$$

Since the circular orbit velocity is

$$v_c = 7.795 \text{ km/sec},$$

then

$$v_1 = v_c + \Delta v_1 = 10.49 \text{ km/sec},$$

where v_1 is the velocity at the end of the third stage. Also since the initial weight of the fourth stage is the burnout weight of the third stage less the structural weight, or M_{i4} = 13,652 - 1096 = 12,556, then

$$\exp \frac{\Delta v_2}{4.116} = \frac{12,556}{M_s + M_l} \tag{14}$$

and

$$v = 10.49 + 4.116 \ln \left[\frac{12,556}{1096 + M_l}\right] \tag{15}$$

is the total velocity required for the two upper stages. The graph for

$$r_\pi \left[= f(v) = h(M_{p4})\right] = g(M_l)$$

is shown in Figure 10. For M_l = 400 pounds r_π = 0.120 A.U., and for M_l = 2500 pounds r_π = 0.245 A.U.

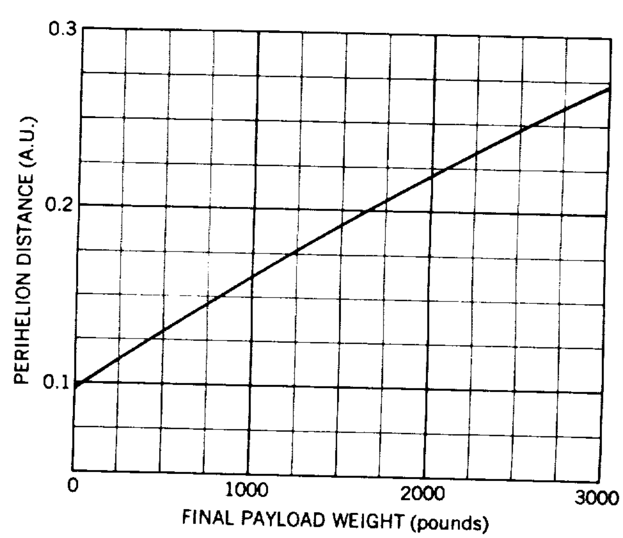

Figure 10—Perihelion distance vs. final payload weight for the two-upper-stage configuration.

Payload and Velocity Capabilities for a Three Stage Vehicle (One Upper Stage)

The total weight of the payload (third stage now), M_c, is $M_p + M_l + M_s$. Since $M_p = 10\,M_s$, $M_c = 11\,M_s + M_l$ or

$$M_s = \frac{M_c - M_l}{11} = \frac{24{,}612 - M_l}{11} \qquad (16)$$

and the rocket equation becomes

$$\exp\frac{\Delta v}{4.116} = \frac{24{,}612}{M_s + M_l} \qquad (17)$$

Figure 11 is a graph of

$$M_s = \alpha(M_l)$$

from Equation 16 and

$$r_\pi \left[= f(v)\right] = \alpha(M_l) = h(M_s)\,.$$

Here, for $M_l = 400$ pounds $r_\pi = 0.185$ A.U., and for $M_l = 2500$ pounds $r_\pi = 0.290$ A.U.

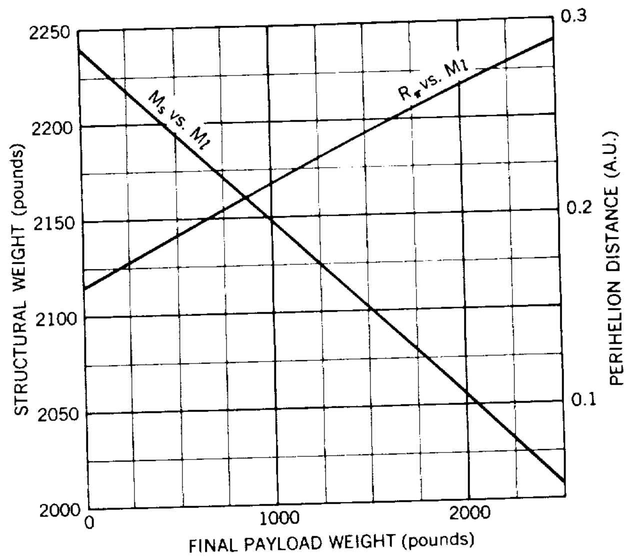

Figure 11—Perihelion distance and structural weight vs. the final payload weight for the three stage configuration.

CONCLUSION

Upon the completion and successful performance of the C-1 rocket and with slight modifications to the specifications of the Centaur Jr., a four stage configuration will be able to orbit payloads of 400 to 2500 pounds at perihelion distances of 0.120 to 0.245 A.U. The minimum perihelion distance attainable for a 1 pound payload is 0.10 A.U.

A more reliable, three stage configuration, having the same specific impulse but requiring greater modifications in structural weight, will be able to deliver payloads of 400 to 2500 pounds to perihelion distances of 0.185 to 0.290 A.U. Its minimum perihelion distance attainable for a 1 pound payload is 0.165 A.U. These figures demonstrate the feasibility of sending useful payloads to the sun.

ACKNOWLEDGMENTS

The author would like to acknowledge Samuel Pines, R. Kenneth Squires, and Ronald Kolenkiewicz for direction and advice in the preparation of this report.

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